AD-A021 472

MODEL 540 MAIN ROTOR BLADE FATIGUE TEST

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Army Air Mobility Research and
Development Laboratory
Fort Eustis, Virginia

January 1976

USAAMRDL-TN-22

070158

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EUSTIS DIRECTORATE
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REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER USAAMRDL-TN-22	2. GOVT ACCESSION NO.	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle) MODEL 540 MAIN ROTOR BLADE FATIGUE TEST		5. TYPE OF REPORT & PERIOD COVERED Technical Note
		6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) Arthur J. Gustafson, Nicholas J. Calapodas		8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, ATTN: SAVDL-EU-TA, Fort Eustis, Virginia 23604		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS 63211A 1F263211DB41 00 002 EK
11. CONTROLLING OFFICE NAME AND ADDRESS		12. REPORT DATE January 1976
		13. NUMBER OF PAGES 39
14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office)		15. SECURITY CLASS. (of this report) Unclassified
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Helicopters
Fatigue
AH-1
Rotor Blade

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The occurrence of catastrophic helicopter accidents due to Bell Model 540-11-250-1 main rotor blade structural failure triggered an Army-wide inspection to determine the condition of the existing 540 blade, and full-scale testing to verify the 1100-hour fatigue life of the Model 540 main rotor blade was conducted at the Eustis Directorate.

Three Model 540 blades, each having varying bond/debond conditions and numbers of accumulated flight hours, were fatigue tested.

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Block 20. Abstract - continued.

The approach taken in this test was to apply a load spectrum the same as that used during the original substantiation of the 540 blade fatigue life conducted by Bell Helicopter Company (BHC).

Existing inspection techniques for bond/debond detection were used and evaluated, with emphasis placed on nondestructive test techniques.

It was concluded that a fully bonded Model 540 blade is flightworthy for 1100 flight hours; blades with accumulated debonds less than 3 feet long are flightworthy for 550 flight hours. However, both surfaces of the blade should be visually inspected before each flight, and they should be ultrasonically inspected every 100 flight hours to determine the debond length. Removal of debonded blades from the inventory is desirable.

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INTRODUCTION

The all-aluminum Bell Model 540 main rotor blade, with a published life of 1100 hours, is used on several helicopter models, including the AH-1G, UH-1C, and UH-1M.

The catastrophic structural failure of four blades after approximately 700 flight hours in service triggered an investigation by an AVSCOM-appointed Risk Assessment Team.

Upon recommendation of the Risk Assessment Team, the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory (USAAMRDL) initiated a program to fatigue test the 540 blade. The test program and its findings are covered in this report.

BACKGROUND

Four Model 540 rotor blade spars developed structural failures in service between 655 and 798 flight hours; two of these were catastrophic failures. Two other blades were found to have cracked skins just aft of the spar after 716 and 867 flight hours.

The structural failures experienced by the four 540 blades were initiated by a debond between the spar and the spar spacer (see Figure 1) that caused local fretting, ultimately resulting in a fatigue crack.

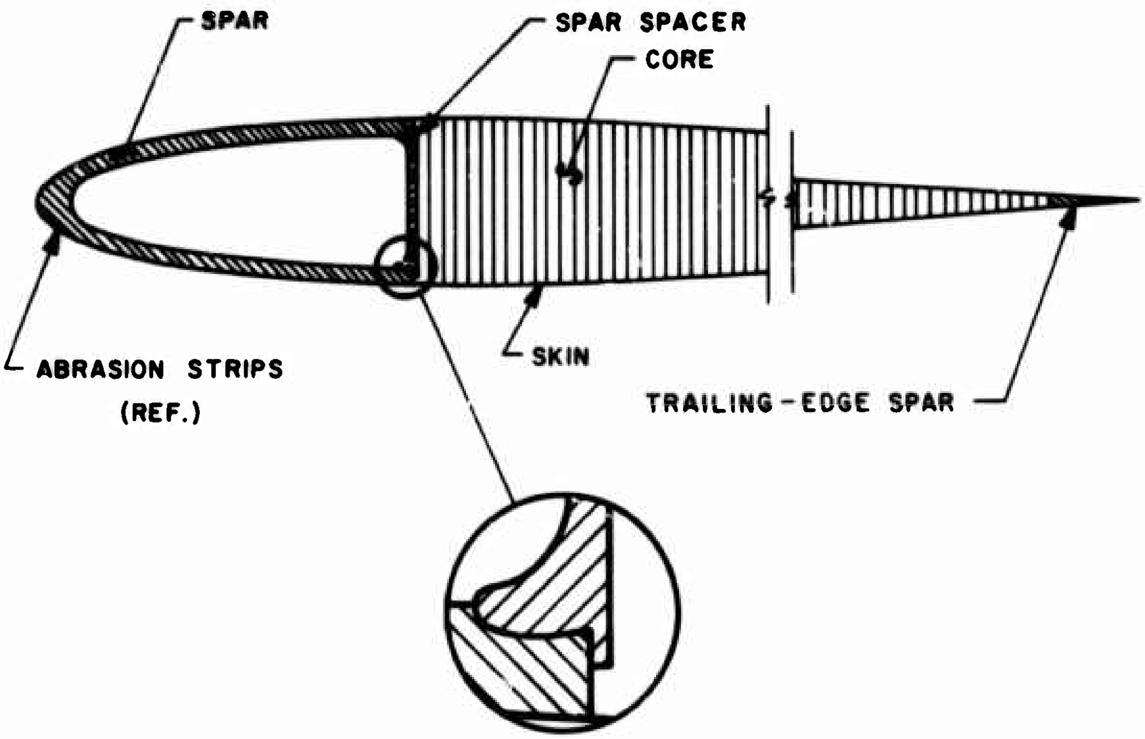

Figure 1. Basic section of the 540 blade.

In response to direction from the Commanding General, U. S. Army Materiel Command, USAAMRDL conducted a technical risk assessment to determine the cause of the Model 540 rotor blade failures on the AH-1 and UH-1 aircraft, and to recommend corrective action. A team of engineering specialists from several Army and NASA agencies conducted the assessment and recommended the following for the Model 540 blade:

1. Inspect all blades with 500 or more flight hours, using the ultrasonic nondestructive testing (NDT) technique currently being developed by AVSCOM's Systems Engineering in conjunction with the Navy at Pensacola. Using the method developed by the Risk Assessment Team, supplement the ultrasonic NDT inspection with a borescope inspection of

the spar. Any spar crack or debond at the closure is a criterion for rejection.

2. Do not fly blades beyond 500 hours (safe life) until the inspection team has been trained and the initial inspection has been conducted.

3. Because the actual sequence of blade degradation is not entirely defined and other causative factors may arise, initially define a 50-hour field inspection frequency, and use the inspection procedures and team defined in 1 and 2 above. (The 50-hour inspection period was computed, based on blade life statistical data, to show a probability of less than one failure in the fleet life, assuming that two opportunities for detecting the incipient failure were provided.)

4. Conduct a subcomponent fatigue test program to establish acceptability criteria for services and/or production defects, and to determine the degradation sequence of the blade.

5. From the results of 4 above, establish a new inspection frequency. It may prove to be feasible to increase the inspection period to 200 hours, which would be approximately one-half of the blade's safe life.

6. Retain the currently published daily inspection procedures.

All of these recommendations were implemented and were in various stages of completion when the 540 blade testing began at the Eustis Directorate. The inspection of all blades with ultrasonics was approximately 80 percent completed, and rejection of blades for spar-to-spar closure debonds ranged between 20 and 50 percent at various Army depots. The high incidence of debonds raised questions concerning acceptable debond length, if any, and the effect of the debonds on loads by a reduction in the torsional frequency. In view of these questions, AVSCOM made the following decisions regarding deficient Model 540 blades:

- Blades from 0 to 550 hours are safe to fly with a total debond length of 36 inches.

- Blades with more than 550 hours with any debonds are unacceptable for flight.

- Blades with 550 to 1100 hours and no debond indications are safe to fly but require a close monitoring inspection program. The inspection frequency is undefined but on the order of 100 hours, starting at 550 hours, and the inspection interval can be predicated on laboratory fatigue test results.

- A recommendation will be made to field commanders that blades with debonds longer than 36 inches be marked with a red X, and that blades with debonds shorter than 36 inches be marked with a circled red X.

FATIGUE TEST PROGRAM OBJECTIVE

The objective of the Eustis Directorate, USAAMRDL 540 rotor blade program was to conduct full-scale fatigue tests of the blade:

1. To determine the mode of bond degradation from an initially sound bonded joint, as defined by the standard ultrasonic inspection technique.

2. To verify the 550-hour safe fatigue life for blades with a 36-inch or less debond.

3. To establish an inspection frequency.

4. To determine the spar fatigue crack growth characteristics of the blade.

5. To evaluate NDT techniques.

TEST SPECIMENS

The blades tested were the Bell Helicopter Company (BHC) design identifications No. 540-011-001-5 and No. 540-011-250-1. The major difference between the two blades is the size of the trailing-edge strip and, consequently, the in-plane stiffness of the blade, with the -250 being the stiffer one.

These blades are approximately 19 feet long with a 27-inch chord and 9-1/3-percent-thick symmetrical airfoil. The spar is made from a C-section extrusion of 2024 aluminum alloy and forms the basic profile of the airfoil leading edge. The spar spacer (see Figure 1) is likewise a 2024 aluminum extrusion and, when bonded in place, controls the airfoil thickness. The lips of the spacer engage the shallow grooves in the inside of the C spar, converting the C section to a D. The tolerance of the spar in its free state and the spacer range from .035 inch clearance to .265 inch interference, excluding bonding adhesives. The blade from the spar aft is composed of aluminum honeycomb and skins bonded to the spar and to an extruded aluminum trailing-edge strip. The spar and trailing-edge strip are tapered chordwise between stations 80 and 140 (see Figure 2); outboard of this point they are of constant chord.

Six blades were selected for the program, and their conditions are presented in Table 1.

These test specimens were modified for testing purposes (see Figures 3 and 4). On a production blade, a lead weight is embedded near the blade tip to increase the local inertia. A 22-inch section from the tip, including the tip weight, was removed to accommodate the end fitting. Also, a laminated stepped aluminum doubler was installed by BHC similar to the one used by BHC in the original fatigue life substantiation of the 540 blade in order to speed the specimen delivery time for minimum design cost.

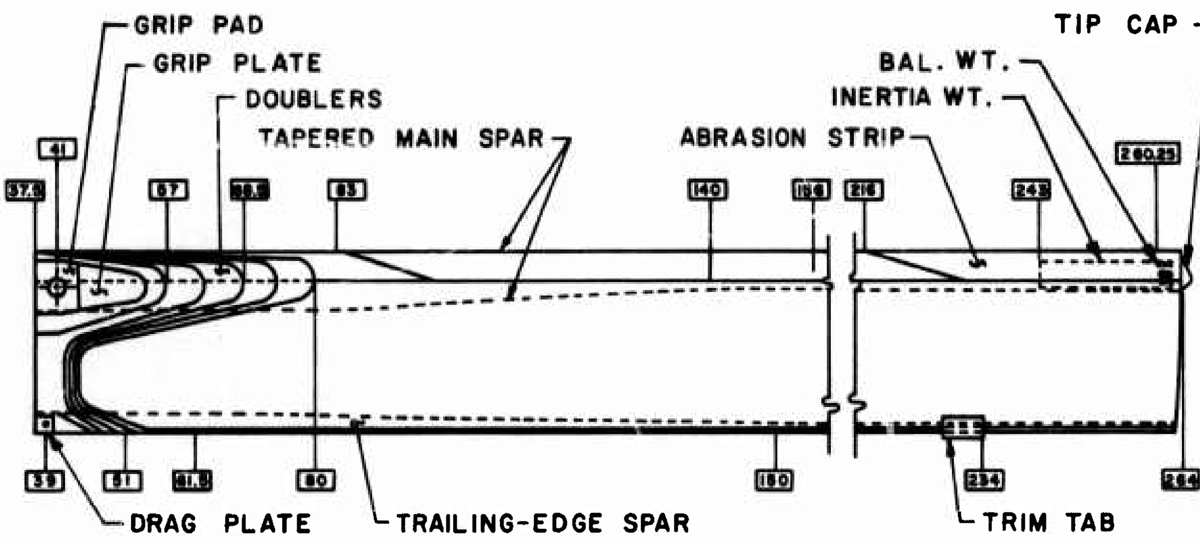

Figure 2. Planform of the 540 blade.

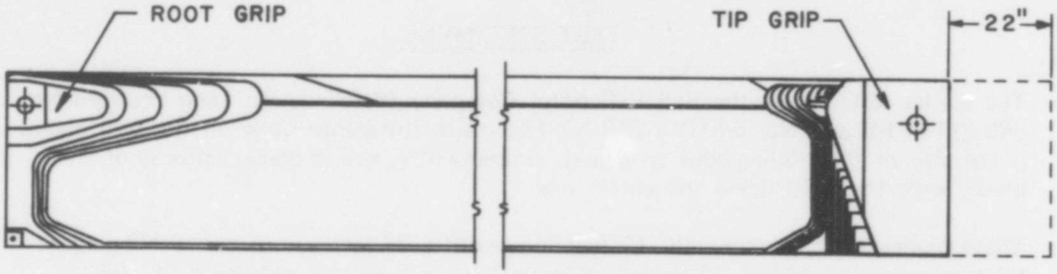

Figure 3. Modified 540 blade for testing.

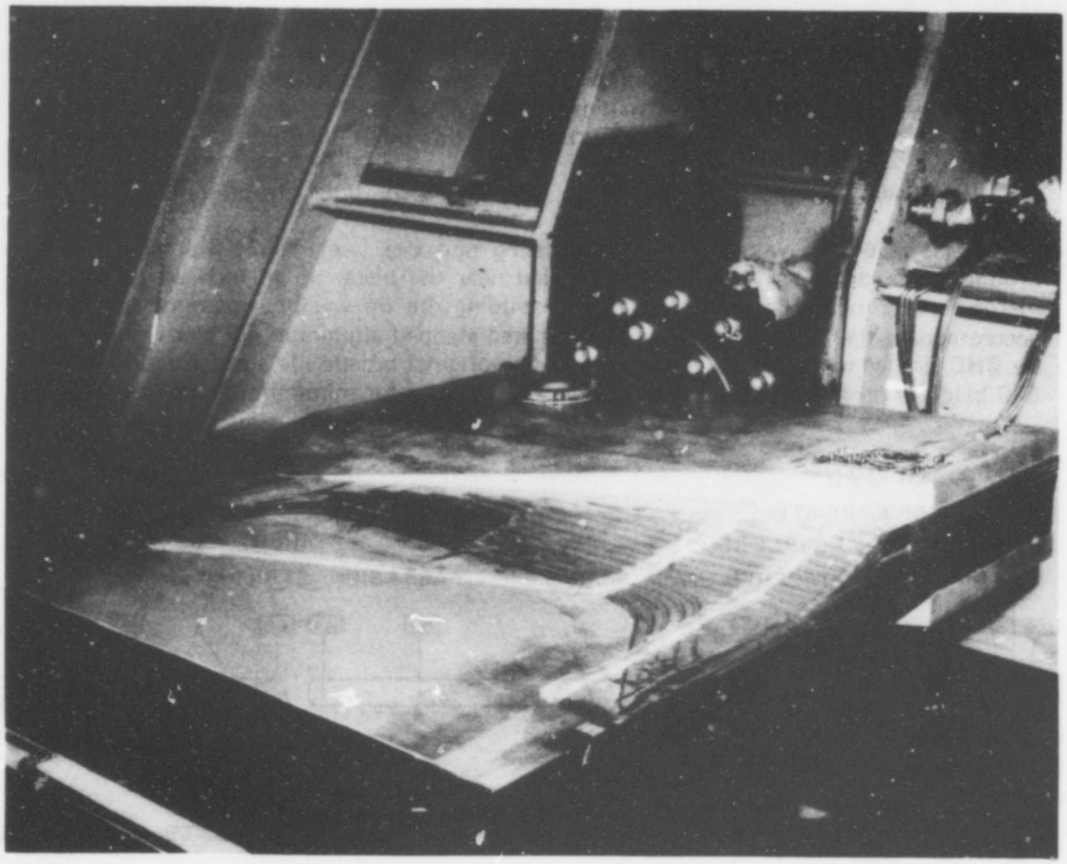

Figure 4. Test specimen with modified tip.

TABLE 1. TEST SPECIMEN CONDITION PRIOR TO TESTING

Serial No.	Debond Station	Flight Hours	Test No.	Blade Type*
IHB 3057	100-112	105	1	-250-1
IHB 3298	no debond	89	2	-250-1
A2-2322	no debond	608	3	-001-5
A2-6462	multiple	0		-001-5
A2-3885**	multiple	217		-250-1
A2-1906	no debond	430		-250-1

*Prefix is 540-011-
**Spare

TEST EQUIPMENT

FATIGUE TESTING MACHINE

The Eustis Directorate, USAAMRDL/Sikorsky 200k-2-108 Fatigue Test Machine (see Figure 5) was used to fatigue test the specimens. This machine is approximately 44 feet long, 10 feet wide, and 12 feet high, and it can accept specimens up to 25 feet long. Initially, it could apply only centrifugal force (CF) and bending loads; it was modified for this test to add torsional loads to its load application capability. The loads were applied on the specimen by two actuators, one located at each end of the test section of the machine.

The machine consists of a frame, a centrifugal loading subsystem, a hydraulic actuator subsystem, and fittings. The centrifugal subsystem can produce an axial force up to 100,000 pounds. The hydraulic actuators for bending and torsion can develop up to 11,000 and 5,000 pounds respectively, and they are limited to displacement strokes of ±2.0 and ±3.0 inches respectively.

The blade was attached on the fatigue machine as shown in Figures 5 and 6.

The tips of all the blades were modified as described in the Test Specimen section. The outboard end of the tip adapter was pinned on the end fitting (see Figure 4), which in turn was pinned to the frame and allowed to rotate on a vertical plane (in the same direction as the flapwise motion of the blade). The end fitting was also pinned on the bending actuator. Four bearing straps, also attached on the end fitting/bending actuator assembly, connected them with the CF shaft and the reaction support rod.

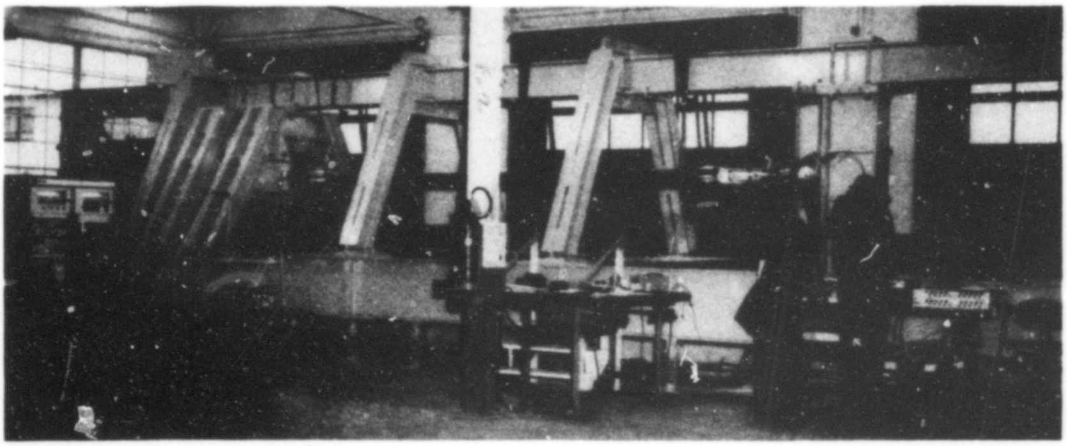

Figure 5. Fatigue machine with 540 specimen attached.

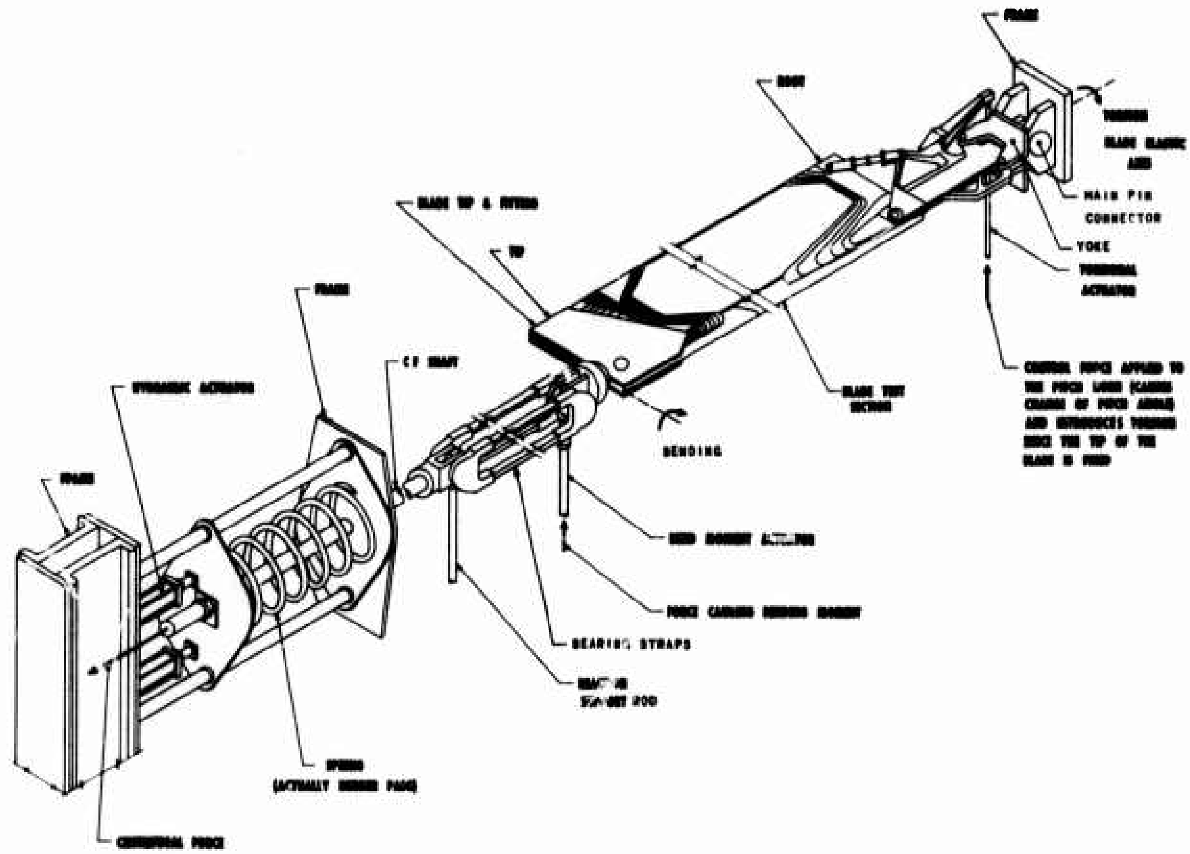

Figure 6. Application of loads.

The reaction support rod (see Figures 6 and 7), acting as a support to the CF shaft, is capable of reacting 100,000 pounds of simulated CF and a bending moment of 2×10^6 in.-lb. The rod itself is pinned on the frame of the machine in a way that allows the rod to rotate only on a vertical plane (same as end fitting). The load simulating the CF was applied on the specimen by pulling on the tension straps using the stack of large rubber pads, which is a soft spring (see Figures 6 and 7). Two hydraulic jacks compressed the rubber pads, and the nut (it can be seen on the left portion of Figures 6 and 7) wrenched to a position that yielded the desired CF, maintaining the compression after the jacks were released. Rubber pads with low spring constant were used in order to minimize the variation in the CF due to dynamic length changes in the specimen.

The hub-root portion remained unchanged for the tests, and the root was installed on the hub as it would be on the aircraft. The hub was attached to a yoke capable of rotating on a vertical plane (see Figures 6 and 8).

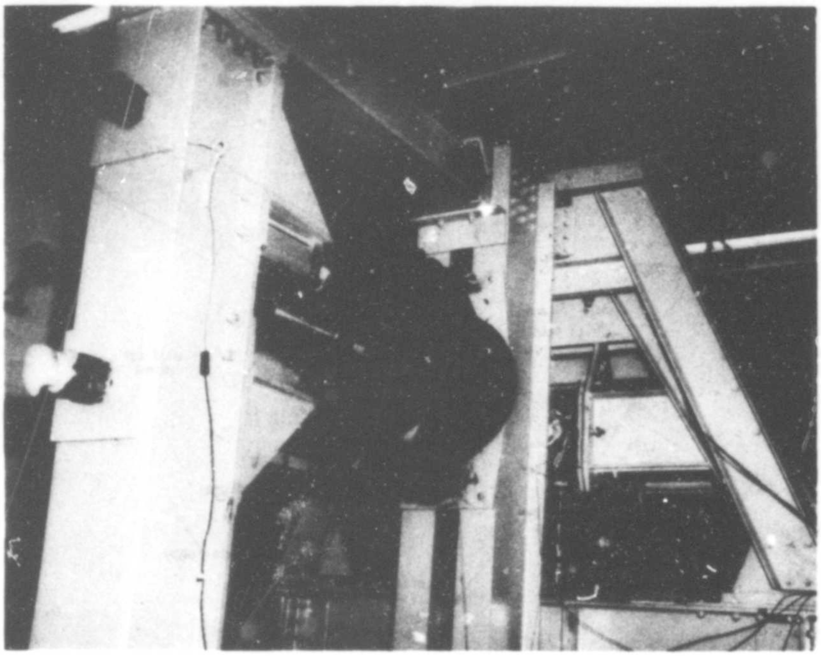

Figure 7. Centrifugal force application subsystem.

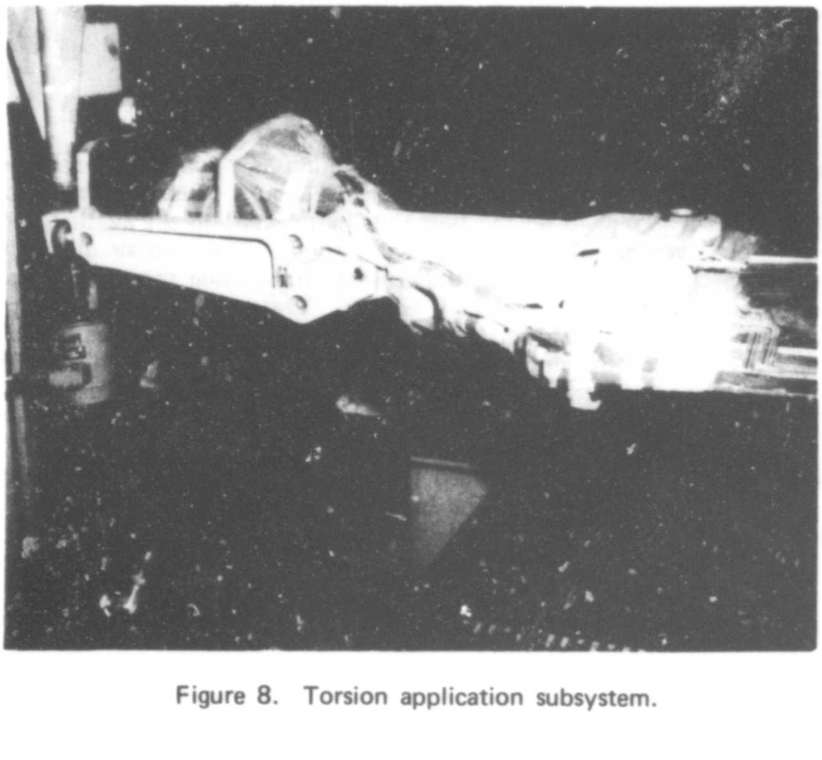

Figure 8. Torsion application subsystem.

The specimen was installed at a pitch angle suitable to avoid large edgewise strain for a given actuator input. The exciting force in the system was provided by a controlled displacement of the electrohydraulic torsion actuator. The torsion actuator (see Figures 6 and 8) is located near the yoke. Its line of action is in the same plane as the main pin connector to lessen flapwise and torque coupling. One end of the actuator is connected to the pitch horn, and the other end is fastened to the structural frame. A load cell in series with the actuator measures the forces being applied.

By controlling the frequency of excitation, the blade-grip system was made to resonate near its first bending mode natural frequency. As the forcing frequency approached the blade-grip natural frequency, the blade's center span amplitude became larger.

LOADS APPLICATION CONTROL SYSTEM

The MTS 810 material test system console was used for the test. It can be seen on the far left side of Figure 5. It consists of the MTS 411.63 arbitrary function generator, MTS 410 digital function generator, MTS 417 center control panel, MTS 422 controller, MTS 413 master control panel, MTS 411.01 data-trak programmer, HP 5326 frequency counter, and MTS 443 controller.

The MTS 411.63 arbitrary function generator was controlled by a 15-load block chart indicating the magnitude and the duration of the load applied to each test blade, and the system translated this information into a torsional actuator displacement.

RECORDERS

The moments and strains developed in the blade due to a torsional actuator displacement were sensed by the proper gages and were recorded by the CEC 5-214 recording oscillograph and CEC 5-133 data graph.

TEST DESCRIPTION

The test consisted of four main tasks: (1) blade instrumentation and calibration of gages; (2) specimen inspection before, during, and after testing; (3) data acquisition; and (4) determination and application of fatigue loads.

BLADE INSTRUMENTATION AND CALIBRATION OF GAGES

Each test blade was instrumented with beam bending, in-plane, and torsion gages as specified in Table 2. Also, strain rosettes were located at appropriate intervals in the region of spar closure debonds.

TABLE 2. SPECIMEN INSTRUMENTATION

	Rosettes		Beam–Chord–Torsion Gage Location		
Test Blade No.	Distance From T.E. of Spar (in.)	Station No.	Distance From L.E. (in.)	Distance From T.E. (in.)	Station No.
1	5.814	84	5.35	.35	85 B, C, T
	4.818	94			110 B, C, T
	3.822	106			135 B, C, T
	2.826	118			160 B, C, T
	1.83	130			
	1.00	155*			
		166*			
		179*			
		191*			
		203*			
2	5.814	80	5.35	.35	110 B, C, T
		93*			135 B, C, T
		105*			160 B, C, T
		117*			185 B, C, T
		129*			213 B, C, T**
	5.70	141			
		153*			
		165*			
		177*			
		189*			
		201*			
		213*			
3		80*	5,35	.35	110 B, C, T
		93*			135 B, C, T
		105*			160 B, C, T
		117*			185 B, C, T
		129*			213 B, C, T**
	5.70	141			
		153*			
		165*			
		177*			
		189*			
		201*			
		213*			

*Added at Eustis Directorate
**Trailing-edge gages added at Eustis Directorate

The rosettes were located on the upper surface of the test blade over the skin with the paint removed, and approximately 1 inch behind the main spar spacer. The choice of this distance was based on two considerations: (1) the torsional rigidity of the sections and (2) the maximum use of rosette sensitivity.

The relationship between the length of a debond and the rosette output is indicated in Figure 9.

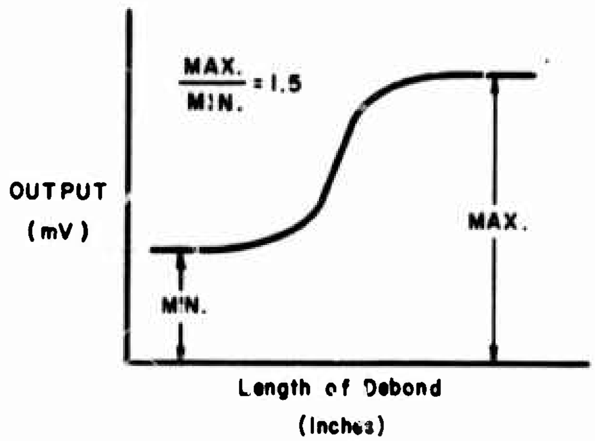

Figure 9. Rosette output and debond length relation.

The ratio between the maximum and minimum output was approximately 1.5. However, the inherent rigidity of a blade varied from one station to another; consequently, the rosette output varied. Therefore, the location of the gages in the chordwise direction at a selected station was such as to assure a comparable strain signal, i.e., maintain the 1.5 ratio, with the strain rosettes distributed at various span stations on the test blades.

The torsional strain component of the rosette was the predominant one in detecting a debond in the spar spacer. In case of a debond, the shear flow within the structural box of the main spar would be interrupted.

The spacer at that station would not carry any torsional loads, the shear would be taken up by the skin, and finally, the shear would be transmitted to the trailing-edge spar. This explains the reduction in shear magnitude indicated in Figure 10.

Had the rosette been located directly over the spacer and had a debond occurred a few stations away from the rosette, the rosette output would have decreased since the spacer would have carried the local torsional load and the skin would have "unloaded".

All blade-mounted strain gages and rosettes were calibrated under static load by Bell Helicopter Company during modifications of the blades for the tests. However, gages were also calibrated by Eustis Directorate, USAAMRDL when gages were replaced.

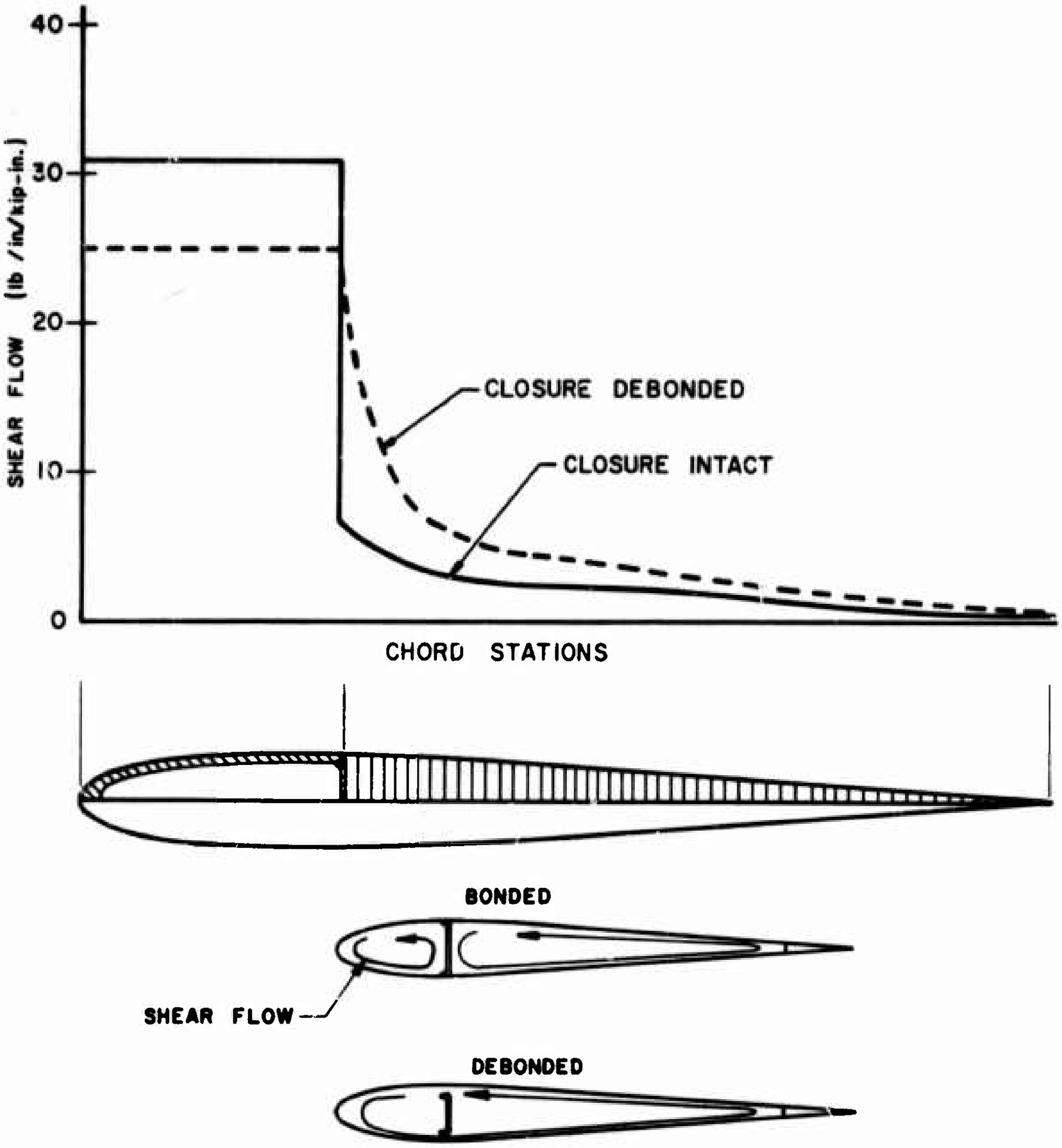

Figure 10. Nominal shear flow distribution at blade station.

Strain gage output was continuously recorded and periodically measured to provide a check of the specimen's condition at chosen stations at a given time and to furnish precise information as to the time, cycle count, extent, and rate of propagation when a debond occurred.

SPECIMEN INSPECTION

Before the test specimen was installed in the fatigue machine, borescopic and ultrasonic tests (NDT) (see Evaluation of Inspection Techniques section) were performed to determine the condition of the blade. During testing, the specimen was continuously monitored visually and ultrasonically inspected at least once a day (i.e., every 990,000 cycles) or as frequently as the situation demanded.

Upon completion of the fatigue testing, the specimen was dismounted and subjected to the following inspection procedure:

- Borescopic and ultrasonic tests were performed.
- The doublers at both ends of the blades were removed.
- The section aft of the main spar, i.e., honeycomb and trailing-edge spar, was removed and the spar was prepared for the pressure test.
- The forward section of the "D" spar was cut in the spanwise direction. The remaining section of the spar (consisting of the spacer and the spanwise strip of the "C" spar with the shallow grooves that engage the lips of the spacer) was cut chordwise at locations where the bonding was intact, thus eliminating the possibility of inducing a debond due to the cutting process. The attached spanwise strip of the "C" spar on the spacer was removed, exposing the debond surfaces.
- The exposed debond surfaces were visually inspected for fretting.

DATA ACQUISITION

The broad aspects of the Model 540 rotor blade fatigue test are described in the Description of Test section. The primary data to be monitored were:

Applied loads
Spar—spar closure bond/debond
Spar cracks
Skin cracks

An operator's log was kept of the following events:

CF load
Start and stop of the fatigue test
Change in load (beam or torsion), observed or induced
Change in span settings (accidental or induced)
Maintenance actions
Identification of active gages

For the above events, the following auxiliary information was recorded:

Name of operator

Time and date
Cycle count
Identification of log entry and recording chart location (sequence number)

In addition to an operator's log, an ultrasonic log was maintained.

DETERMINATION AND APPLICATION OF FATIGUE LOADS

The loads used for the test were furnished by BHC and were identical to the load spectrum used by BHC in its original fatigue life substantiation of the Model 540 blade. The in-plane loading (due to drag force) has a negligible effect on the spar closure fatigue life; therefore, it was not applied.

The end conditions of the blade during test (pinned–pinned) were different from those in actual flight (pinned–free), resulting in a frequency of vibration of the blade during testing of approximately twice the in-flight load frequency. The frequency of vibration was determined by the natural bending frequency of the blade-grip assembly.

The load spectrum and its rate of application were considered to be important parameters in this test. BHC provided flight load data that shows the interrelationship of the bending and torsional loads as well as the wave shape of each. Figure 11 indicates the trace of loads applied during flight and during fatigue testing.

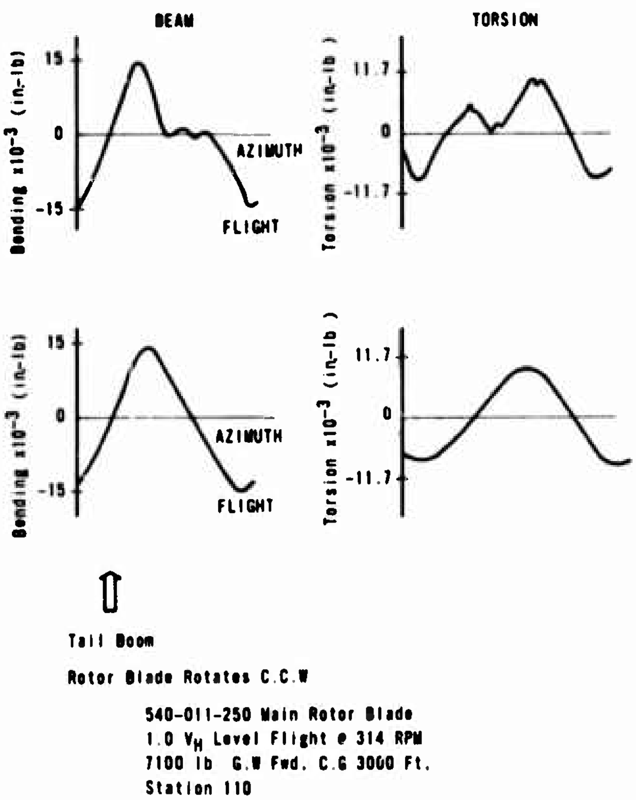

Figure 11. Representative load trace on the 540 blade.

The load spectrum furnished by BHC represented approximately one flight hour, and included 15 different blocks of load segments encompassing the entire beam and torsion spectrum.

The centrifugal force is a body force, and its magnitude is a function of the blade length, the mass distribution along the blade, and the rpm. In the fatigue test, the flight loads furnished by BHC were represented by a set of external loads, and the CF was represented by a uniform tensile load. Since the simulated CF load is uniform along the full length of the specimen and since the in-flight CF varies with blade station, the simulated loading is equal to the flight loads at one station only. Consequently, the simulated loads were calculated to be equal to the flight loads at a chosen blade section.

The test section on the first specimen was taken at station 110, which was the midpoint of an existing debond. Station 147, i.e., the midpoint of the blade, was the test section on the remaining two specimens. The loads for station 147 were obtained by interpolating the loads and the percentage of their occurrence between stations 135 and 160; the information for stations 135 and 160 was furnished by BHC. Tables 3 and 4 indicate the loads imposed on the test blades at stations 147 and 110 respectively. Table 5 indicates the randomly selected sequence of the applied load blocks during the testing.

TABLE 3. FLIGHT HOUR LOAD SCHEDULE FOR STATION 147
(Blades IHB 3298 and A2-2822)

Block No	Alternating Torsion Moment (in.-lb)*	Alternating Beam Moment (in.-lb)	No. of Load Cycles/Flt Hr**
1	707	1,000	270
2	1,410	2,000	275
3	2,120	3,000	545
4	2,530	4,000	965
5	3,540	5,000	2,880
6	4,240	6,000	6,170
7	4,950	7,000	4,392
8	5,660	8,000	907
9	6,360	9,000	900
10	7,070	10,000	365
11	7,777	11,000	275
12	5,450	12,000	270
13	9,190	13,000	416
14	9,900	14,000	270
15	10,600	15,000	540

*Based on a torsion-to-beam load ratio of .717
**Based on 325 rpm

TABLE 4. FLIGHT HOUR LOAD SCHEDULE FOR STATION 110
(Blade IHB 3057).

Block No.	Alternating Torsion Moment (in.-lb)*	Alternating Beam Moment (in.-lb)	No. of Load Cycles/Flt Hr**
1	779	1,000	195.0
2	1,558	2,000	429.0
3	2,337	3,000	390.0
4	3,116	4,000	2,047.5
5	3,895	5,000	1,195.0
6	4,674	6,000	3,510.0
7	5,453	7,000	5,070.0
8	6,232	8,000	2,535.0
9	7,011	9,000	2,486.25
10	7,790	10,000	390.0
11	8,569	11,000	243.75
12	9,348	12,000	146.25
13	10,127	13,000	146.25
14	10,906	14,000	0
15	11,685	15,000	48.75

*Based on a torsion-to-beam load ratio of .779
**Based on 325 rpm

NOTE: Bending and CF loads were set for their values at the midpoint of the debond. For example, for Blade IHB 3057, the CF is 87K. The phase angle between beam and torsional loads is 74°, torsion lagging. The steady moment loads are: Beam = 5000 in.-lb and Torsion = 8000 in.-lb.

TABLE 5. TEST LOAD BLOCKS APPLICATION SEQUENCE
(Random)

Sequence	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Block	4	9	10	12	3	6	7	1	8	14	11	5	2	15	13

Difficulty was experienced in applying the correct loads on the test blades. The fatigue machine, although massive, proved to be sensitive to temperature changes. Thermocouples were located at appropriate points on the machine–blade system, and the effect of temperature on various components was examined. On the average, a 5° to 8°F change in temperature resulted in approximately 1000-1500 pounds change in the CF.

This problem was compounded by drift in torsional actuator frequency. Precautions were taken to operate on the lower slope region of the respective frequency curve, slightly under resonance point, in order to avoid large changes in amplitude for small changes of frequency.

Due to the lack of climatic control, maintaining the correct loads in the machine—blade—electronic equipment system was somewhat difficult because of the unpredictable behavior of the components due to temperature and frequency variations; thus the operators had to adjust the torsional frequency setting frequently in order to maintain the CF constant. As a result of careful attention to this point, the difficulty in maintaining the loads did not affect the accuracy of the results.

The application of correct loads was aided by the establishment of operating boundaries using strain gage responses during sequences 6 and 13.

Sequence 6, due to its long duration (approximately 10 minutes) and relatively low load, was used in performing frequency adjustments to obtain the desired torsion-to-bending ratio. It was important to obtain the correct ratio particularly in sequence 13 since it corresponded to the highest load block applied on the test blade.

The torsion-to-bending ratio was extracted from torsion and bending moment in-flight data (furnished by BHC), such as that shown in Figure 12. The slope-defining line was drawn through the highest concentration of the scatter of the plotted points using visual regression. The resulting slope is later normalized by multiplying it times the maximum torsion to maximum bending ratio, and that constitutes the desired torsion-to-bending ratio used for the test. Obviously, the choice of such torsion-to-bending ratio makes small variations from it acceptable; consequently, their causative loads are acceptable.

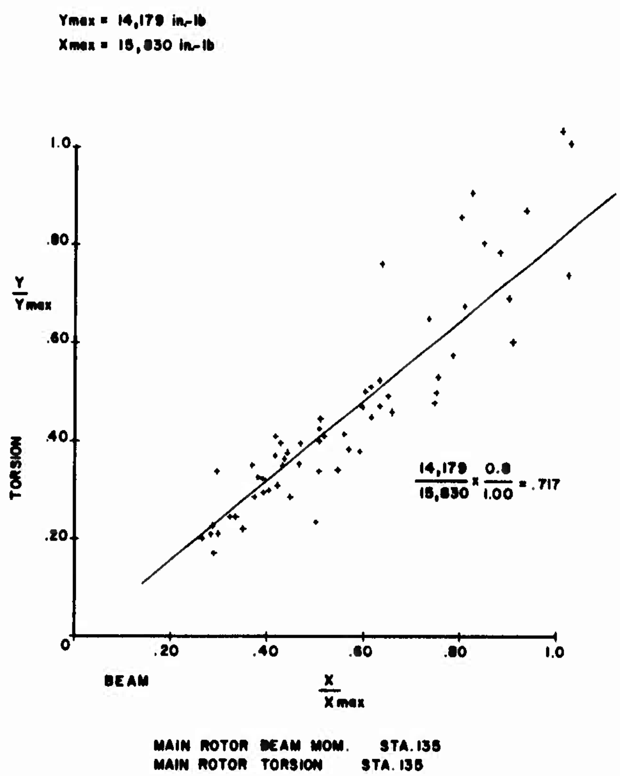

Figure 12. Oscillatory beam-to-torsion relation of the 540 blade at a given station.

TEST RESULTS

Using a fatigue loading spectrum identical to the one used by BHC in its original fatigue life substantiation of the 540 blade (see BHC report 309-099-06), the Eustis Directorate, USAAMRDL 540 fatigue test program was completed.

Initially, six blades were selected for the test (for more details, see Test Specimen section): five for testing and one for a spare. The tip sections of all the blades had to be modified in order to mount the blades on the fatigue machine.

During the test, it became evident that the essential objectives of the test could be accomplished by testing only three blades. The first, second, and third blades were tested for 550 simulated flight hours each (about 10,692,280 cycles). Upon completion of the third blade testing, the second blade was remounted on the fatigue machine and tested for an additional 550 hours. Thus, three test blades were used, but the 540 blade fatigue life determination was actually based on four simulated flight hour blocks.

The results of these tests are summarized below.

TEST BLADE NO. 1

The first blade tested was the IHB 3057, which had 105 service flight hours and an existing debond between stations 102 and 114 (12 inches). After accumulating 3,402,000 fatigue cycles, the debond began to propagate. At 3,985,200 cycles, the debond had extended from station 10 to station 142, representing a total extension rate of 1.67 inches per simulated flight hour. The debond at this point slowed down and extended only 8.5 inches during the next 5,905,800 cycles.

At 9,891,000 fatigue cycles, a portion of the trailing-edge box and the spar closure at station 212 failed in fatigue. Figures 13 and 14 indicate the failure in the upper and lower surface of the blade respectively. This failure was induced by the end fitting and the fact that all loads outboard of station 110 (which is the center of the test section) were higher than the fatigue spectrum loads. The failure was not a result of the spar-to-spar closure debond. No damage was experienced by the spar, and the blade was capable of reacting the static and alternating loads. Consequently, the trailing-edge box was patched as shown in Figure 15, and an additional 40,000 fatigue cycles were run to confirm that the applied loads in the region of the debond were equal to the loads prior to failure. The loads inboard of station 160 were found to be the same as before the failure at station 212. The test was terminated on this blade, as further testing would have resulted in failure at the tip grip.

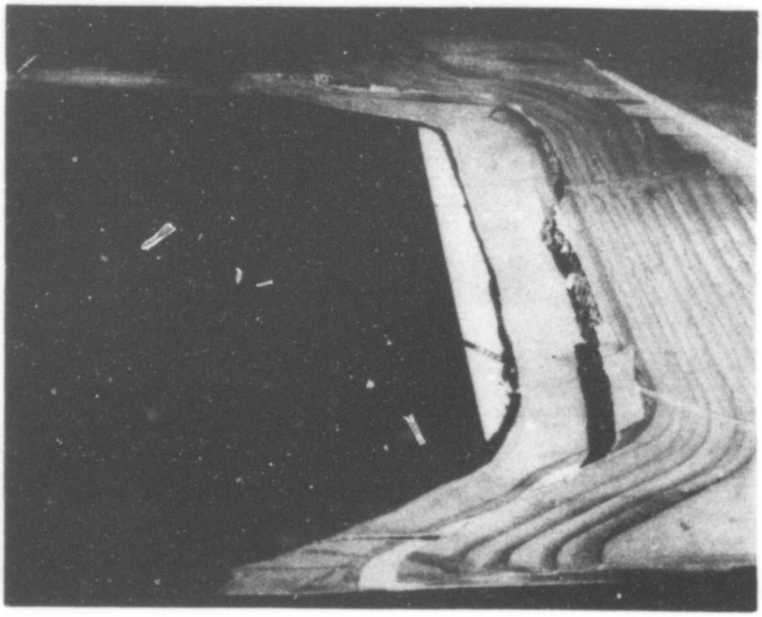

Figure 13. View of trailing-edge box failure near the tip, top surface, test specimen No. 1.

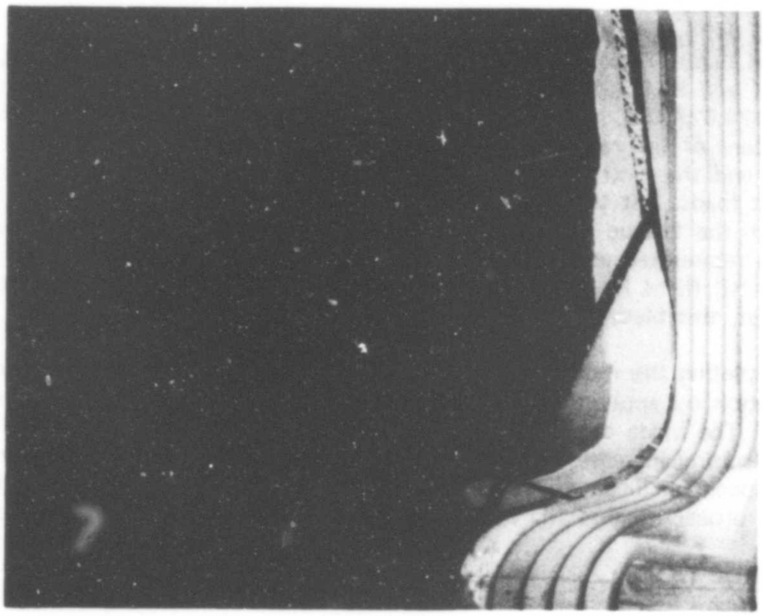

Figure 14. View of trailing-edge box failure near the tip, bottom surface, test specimen No. 1.

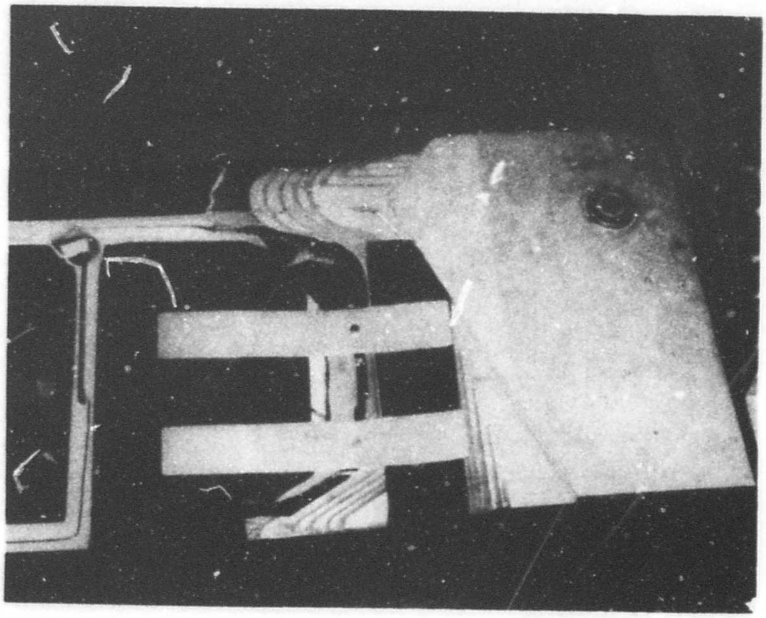

Figure 15. Braced section of test specimen No. 1.

TEST BLADE NO. 2

The second blade, IHB 3298, had 89 service flight hours and an existing debond between stations 81 and 83.5 (2.5 inches). The total number of fatigue cycles imposed on the blade was 24,627,480. However, 3,243,100 of those cycles were applied with only 50 to 68 percent of the recommended beam bending loads. This discrepancy was corrected, and the next 10,692 cycles or 550 simulated flight hours were applied with the correct loads. At this point, the blade was removed and test blade No. 3 was installed on the fatigue machine. When tests of blade No. 3 were concluded, blade No. 2 was remounted and tested for an additional 10,692,100 cycles or approximately 550 simulated flight hours, thus constituting the fourth 550-flight-hour block. Throughout the test, this blade experienced no further debond.

After completing the requirement for 1100 simulated flight hours, the blade was fatigued to destruction by applying gradually increasing loads greatly beyond the accepted load spectrum. The blade debonded between stations 80 and 123 and stations 200 and 213, and the resulting increase in strain was recorded. The objective was to provide a damaged specimen in an effort to evaluate the nondestructive test techniques (NDT). The inspection procedure that the blade underwent is described in the Test Description section.

TEST BLADE NO. 3

The third blade was the A2-2822, which had 608 service flight hours and no debonds. After 10,692,100 cycles or 550 simulated flight hours, there was no sign of bond degradation.

EVALUATION OF INSPECTION TECHNIQUES

The techniques evaluated were the same as those used on the 540 rotor blade program by various agencies in their efforts to develop an effective spar inspection technique.

Table 6 indicates the inspection techniques used on each test specimen.

TABLE 6. INSPECTION TECHNIQUES

Blade No.	Test Specimen	Nondestructive: Ultrasonic	Nondestructive: Borescopic	Destructive: Pressure	Destructive: Red Dye	Destructive: Ultraviolet	Destructive: Visual
1	IHB 3057	X	X	X	X	X	X
2	IHB 3298	X	X	X			X
3	A2-2822	X	X	X			X

ULTRASONIC TECHNIQUE

This technique consists of ultrasonic through-transmission and ultrasonic shear-wave transmission. The equipment used was the Sperry Model UM 721 reflectoscope with a Model 5 NRF pulser-receiver unit and 5MHz transducers.

The ultrasonic through-transmission technique uses two transducers: the ultrasonic transmitter and the ultrasonic receiver. The transmitter transducer is placed on one side of the test specimen, and the receiver is placed on the other side of the test specimen directly opposite the transmitter transducer. The ultrasonic signal is transmitted through the test specimen to the receiver, and both the transmitted signal and the received signal are displayed on the oscilloscope. Lack of the received signal indicates a discontinuity in the test specimen.

The ultrasonic shear-wave transmission uses one transducer for both the transmitter and the receiver. The ultrasonic signal is transmtted at an angle into the test specimen. If a discontinuity, such as a crack or edge of the test specimen, is in the path of the signal, the signal is reflected back to the transducer and is displayed on the oscilloscope.

The through-transmission technique was used to inspect the 540 test blades for debonds. With the exception of test blade IHB 3057 (No. 1), the paint over both sides of the trailing edge of the main spar was removed prior to the initial ultrasonic inspection, and the location of the spacer was marked on the top skin of the blade from stations 80 through 213 (see Figure 2). This was done to ensure the proper positioning of the transducers over the adhesive bond for the top skin surface of the blade over the main spar spacer, and the other transducer was placed on the bottom skin surface over the opposite end of the spacer.

All test blades were inspected upon receipt, both ultrasonically and borescopically, which indicated a good correlation with the data received from BHC.

The information on the bond conditions of the specimen obtained from the ultrasonic through-transmission are depicted in Figures 16 through 19. Some stations could not be inspected due to the strain gage location. Those locations are represented by the dotted lines.

As can be seen from Figures 16 through 19, the ultrasonic signal is proportional to the alleged quality of the bond. Consequently, the quality of the bond needs to be determined before a legitimate correlation of the ultrasonic signal and bond quality, if any, can be established. To establish this correlation, upon completion of fatigue testing and post inspection (with the exception of visual test), the main spar was cut as outlined in the Test Description section. Figure 20 shows a sample of the main spar sections that were used for the above correlation.

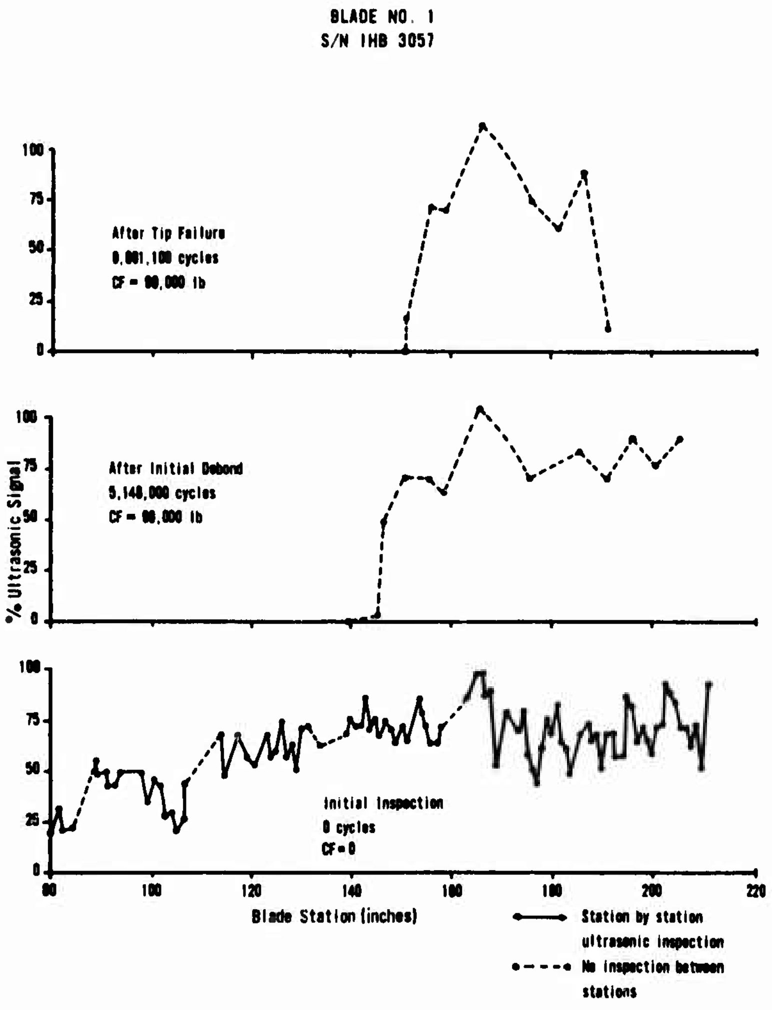

Figure 16. Ultrasonic histogram of test specimen No. 1.

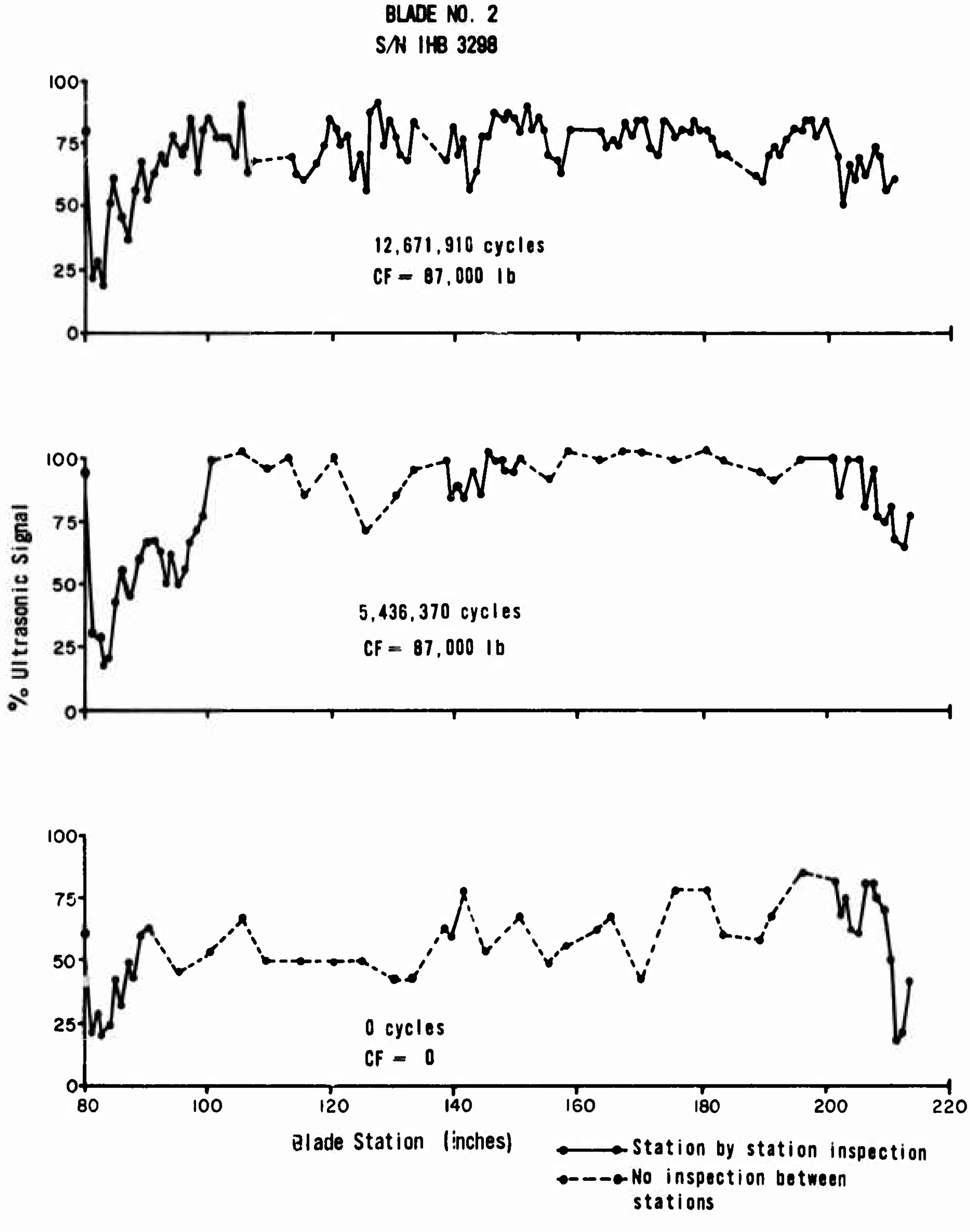

Figure 17. Ultrasonic histogram of test specimen No. 2.

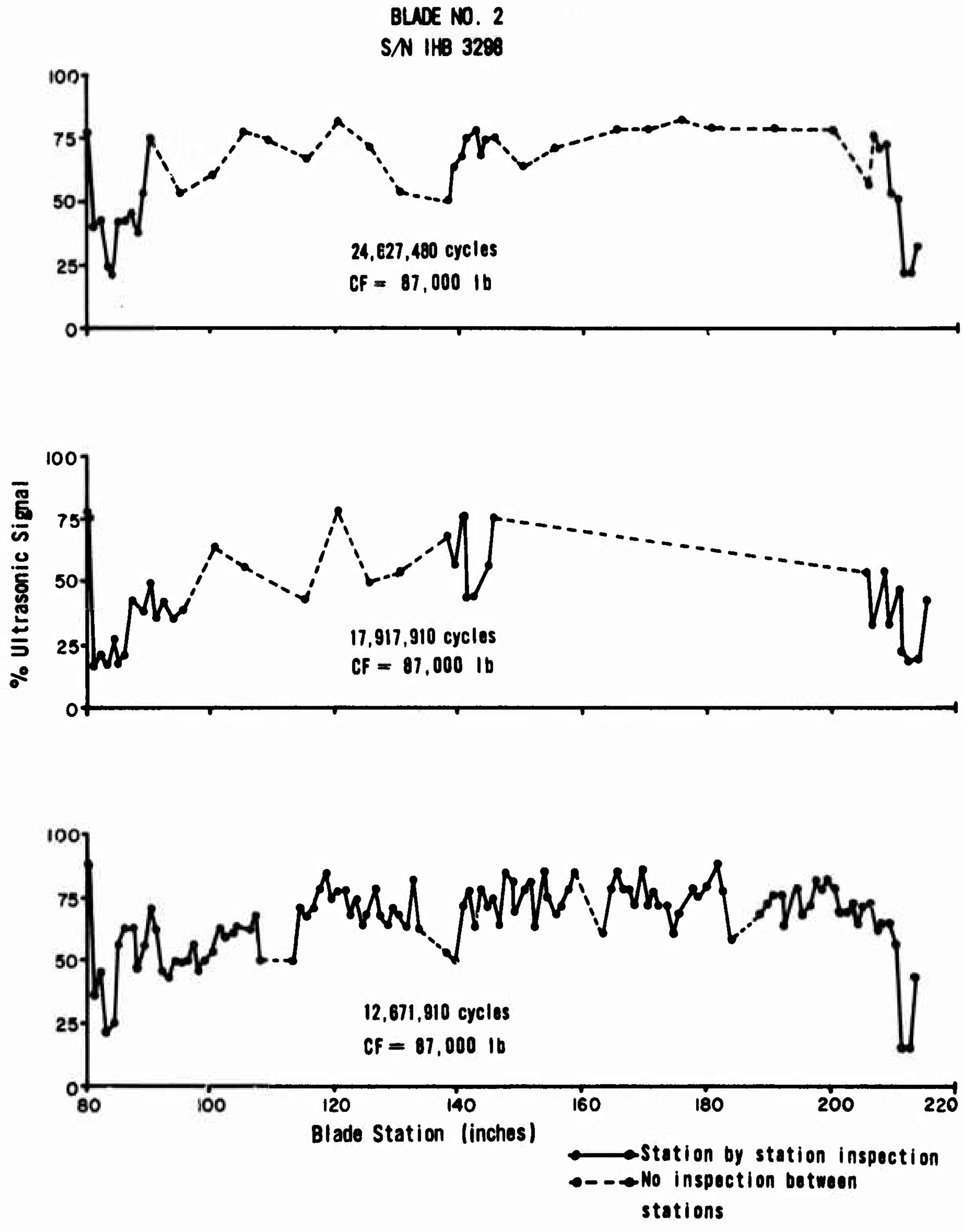

Figure 18. Ultrasonic histogram of test specimen No. 2B.

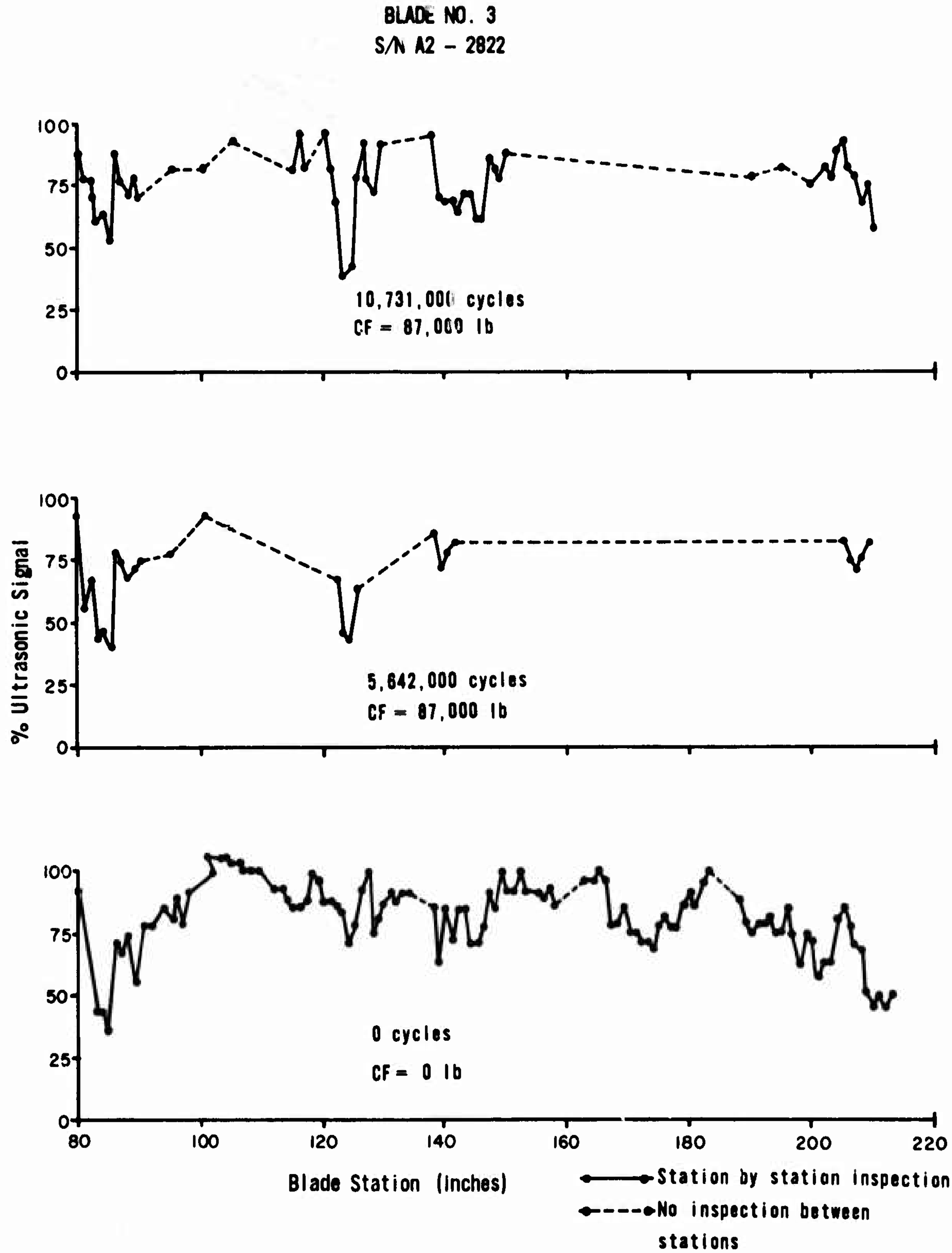

Figure 19. Ultrasonic histogram of test specimen No. 3.

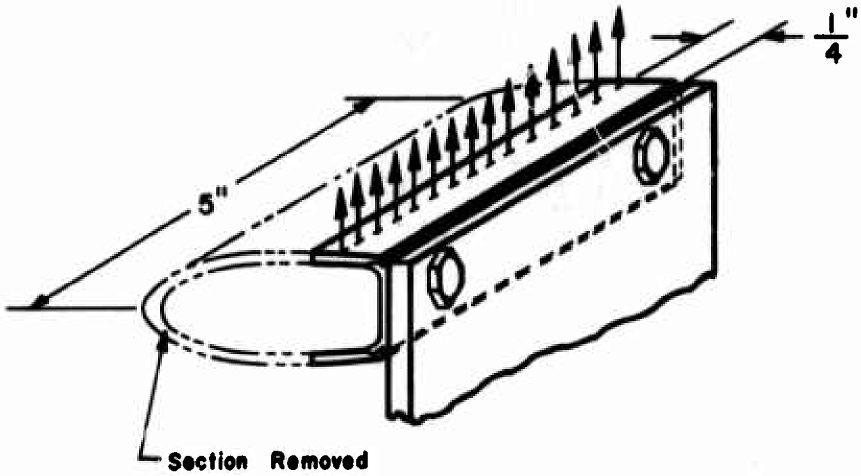

Figure 20. Bond quality inspection specimen.

Five-inch-long specimens were chosen to provide an adequate bond length free of any induced debonds that could have been initiated during the cutting process, and they correspond to high, medium, and low ultrasonic reading areas. A force applicator was fabricated to apply the force uniformly. The required bending moment to disassemble the specimens (see Table 2) was recorded and compared with the ultrasonic signal previously obtained at the respective stations (Table 7). An average ultrasonic signal intensity was used, and it was obtained over the 5-inch specimen.

TABLE 7. CORRELATION OF ULTRASONIC SIGNAL AND BENDING MOMENT

Blade	Station	Side	Force (lb)	Moment Arm (in.)	Moment (in.-lb)	% Ultrasonic Signal
2	100	Top	1643	.25	411	-
2	130	Top	1411		363	80
2	130	Bottom	-		-	-
2	160	Top	1929		482	60
2	160	Bottom	1169		292	60
3	85	Top	2139		535	92
3	85	Bottom	1863		466	92
3	145	Top	1096		274	83
3	145	Bottom	1863		466	83
3	157	Top	1632		408	87
3	157	Bottom	1731		433	87
3	163	Top	1654		414	88
3	163	Bottom	1731		433	88
3	210	Top	1058		264	68
3	210	Bottom	1907		477	68

Based on the results in Table 7, it appears that firm correlation of the ultrasonic signal with the bond quality cannot be made. Additional testing will be necessary. However, the ultrasonic signal will give the inspector a conservative estimate of the bond quality, and passing a blade with an unacceptable debond length is unlikely.

The evaluation of the ultrasonic NDT technique is summarized as follows:

- Of all the inspection methods used for the tests, this is the only one that can be performed without disassembling the blade. All ultrasonic testing in the 540 program was done on test specimens off the aircraft; however, tests conducted by other Army agencies established that this technique can be used with the blade on the aircraft.

- The ultrasonic technique, in most cases, can indicate the condition of a bond; i.e., complete or partial debond.

- Personnel can be quickly trained to handle the ultrasonic test equipment and to conduct the inspection.

- Operating cost is low.

BORESCOPIC TECHNIQUE

The borescope used was the American Cystoscope Makers, Inc., Model No. B-7536A. The test specimen underwent borescopic inspection before it was mounted on the fatigue machine for the test and after it was dismounted upon completion of the test. The spar end seal of the specimen had to be removed to allow insertion of the borescope.

This inspection method relies strictly on visual observation, and had a debond been concealed by excess adhesive, it would not have been detected.

Figures 21 through 23 indicate cracks as they can be seen through a borescope. The first two figures correspond to cracks in the outermost fibers of the excess adhesive adjacent to a sound bond. The last figure corresponds to an all-the-way-through crack. It is obvious that it is almost impossible to identify the "real" crack and that the use of an alternate inspection technique is mandatory. Ultrasonic through-transmission was used to detect the extent of the crack.

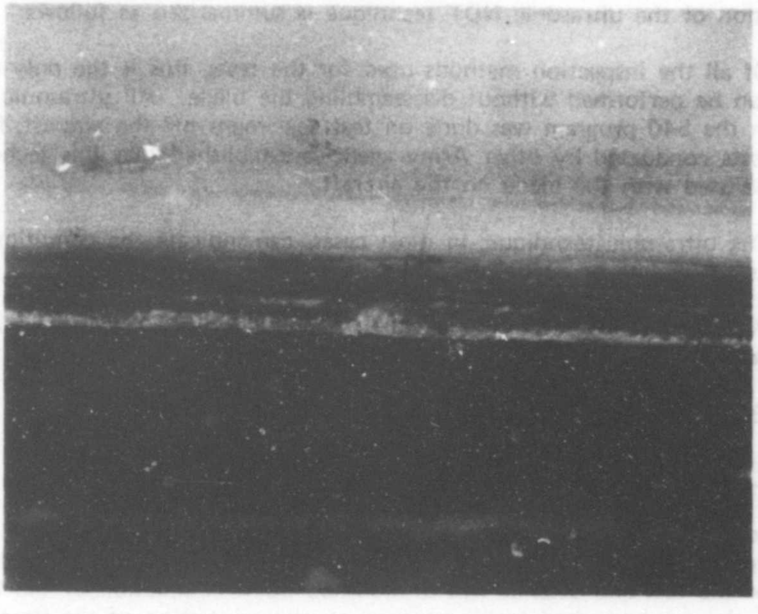

Figure 21. Crack existing on the bondline excess adhesive viewed through a borescope.

Figure 22. Crack existing on the bondline excess adhesive viewed through a borescope.

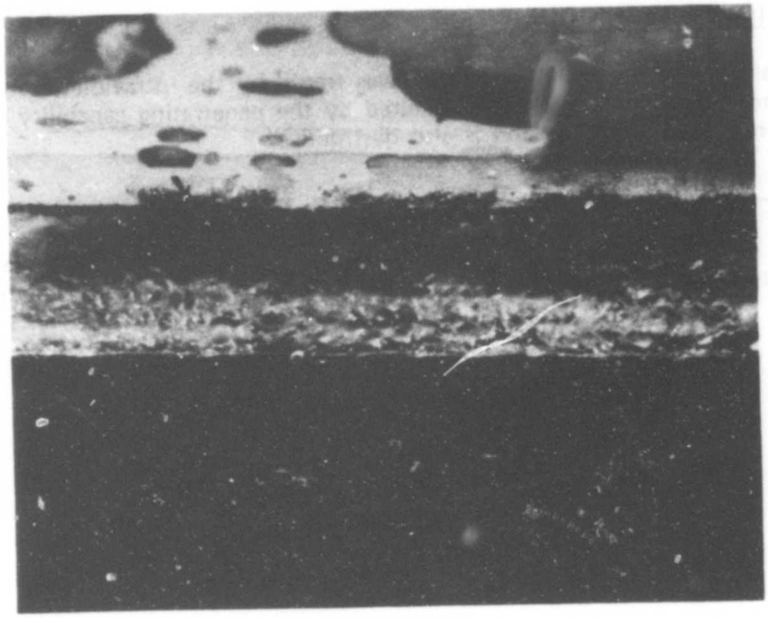

Figure 23. All-the-way-through crack viewed through a borescope.

PRESSURE TECHNIQUE

This inspection method used pressurized air for debond detection. Two aluminum seals were fabricated and were installed inside the spar ends. Also, the honeycomb adjacent to the spacer had to be removed. Then the spar was pressurized at approximately 10 psig, and the spacer was covered with soapy water. Had a debond existed, the escaping air would have caused the solution to bubble. However, a debond sealed by excess adhesive would be concealed.

This technique can be applied only during rotor blade repairing or overhauling.

RED DYE TECHNIQUE

This method consisted of gravity feeding a red dye/alcohol solution into the interior of the spar along the bond lines. As with the borescopic and pressure techniques, this method will not indicate a debond if the debond is concealed by excess adhesive. It is also a destructive test.

ULTRAVIOLET TECHNIQUE

The penetration of the red dye solution was traced by the ultraviolet test, which is an effective test in itself; however, it is limited by the penetrating capability of the fluorescent solution (red dye). This method is also destructive.

VISUAL TECHNIQUE

As stated in the Ultrasonics section, the 5-inch specimens were disassembled and the bond line was inspected microscopically. Weak bond areas could be easily identified.

CONCLUSIONS

Based on the results of the Model 540 Rotor Blade Fatigue Test Program, it is concluded that:

1. Fully bonded blades are flightworthy for 1100 flight hours.

2. Blades with accumulated debonds totaling less than 3 feet in length are flightworthy for 550 flight hours. However, the blade top surface should be visually inspected before each flight and ultrasonically inspected every 100 flight hours. These blades should be removed from the inventory as soon as possible.

3. Ultrasonic inspection is an adequate nondestructive test technique for detecting debonds.